

II. Viking Project, Orbiter System and Project Support

A. Project Description and Status

1. Description

The primary objective of the *Viking* Project is to send two vehicles to the planet Mars to perform scientific experiments directed toward enhancing current knowledge about the physical characteristics of the planet, particularly its capability for supporting life and possible evidence of life. The two vehicles, each consisting of an orbiter system and a lander system, are anticipated for launch during 1975. The orbiter system will be developed by JPL, and the lander system will be developed by the Martin-Marietta Corp. Langley Research Center has overall management responsibility for the *Viking* Project.

The orbiter system will transport and inject the lander system at the appropriate point for a selected landing site and will relay telemetered data from the lander to earth. Scientific instruments on the orbiter will be used to measure atmospheric and surface parameters at various times and locations to determine the dynamic characteristics of the planet. The topography of Mars will be

mapped during orbital operations, with special emphasis on mapping the proposed landing site prior to deorbit of the lander system and on supporting the lander after it is on the surface of Mars by determining if changes are apparent in the vicinity of the lander. Both visual and infrared coverage will be possible during the presently planned 140 days of orbital operations.

During entry and after landing, the scientific instruments on the lander will measure Mars' atmospheric composition, temperature, pressure, and density. After landing, the topography of the landing site will be mapped, and measurements will be made of the planet's surface composition, temperature, pressure, humidity, and wind speed. Of particular interest in the surface measurements are the type of organic compounds present, if any, and the amount and form of water. A gas chromatograph/mass spectrometer, for measuring both atmospheric and surface composition, is being developed by JPL. The surface soil analyses will be directed at detecting evidence of growth and/or metabolism.

2. Status

JPL's cost estimates for the reprogrammed *Viking* orbiter 1975 activities were presented to the Project Office at Langley Research Center and were followed by an on-site review of cost backup data at JPL.

Work is essentially completed on commonality studies between the Venus-Mercury 1973 and the *Viking* orbiter 1975 missions. However, the changes in implementation modes that are necessary to achieve the savings possible through commonality, plus the requirement for early commitment of resources and design requirements, may make commonality difficult to utilize. Additional review effort is currently aimed at determining if some subsystems should be acquired in a common arrangement, with benefit to both projects.

B. Guidance and Control

1. Automatic Sun Occultation—Sun Acquisition Control for the *Viking* Spacecraft

a. Introduction. The *Viking* spacecraft orbital trajectory is such that the planet will pass between the spacecraft and the sun once each orbit during part of the orbital phase of the mission. The time duration of the sun occultation will extend from a few minutes to as much as 2 h, and to maintain attitude control during this period, the gyros will control the spacecraft in the inertial mode of operation. A proposed method to achieve this mode of operation is to detect occultation through the use of the sun gate in conjunction with the acquisition and cruise sun sensors. A signal from these sensors will be used to initiate inertial control of the spacecraft.

b. Mechanization. Automatic sun occultation control would be performed on the *Viking* spacecraft by detecting the difference between the sun occultation mode and the sun acquisition mode. This information would then be used to place the gyros either in rate mode (sun acquisition) or in inertial mode (sun occultation). This mechanization works in the following manner for sun occultation: As the spacecraft begins sun occultation, the dark threshold of the sun gate is exceeded. This causes the sun gate electronics to change state, which turns on the gyros. At the same time, the sun sensors are being monitored by a level detect amplifier. Since there is no sun in the field of view of the sensors and the dark output impedance of the sun sensors is extremely high ($>20\text{ M}\Omega$), the output will be at null. This information is used by the sun sensor level detect amplifier to put

the gyros in the inertial mode. The gyros stay in inertial hold until the sun reappears in the sun gate field of view after sun occultation. At this time, the gyros will be turned off and the spacecraft returned to celestial control (sun and Canopus).

This method of occultation control has the advantage that it is automatic (requires no switching—no central computer and sequencer interface) and can always distinguish between sun acquisition and sun occultation.

For the sun acquisition mode, this mechanization detects the loss of the sun gate and when the sun is in the field of view of either the acquisition sun sensors or the cruise sensors. The voltage errors out of the sensors are continuously monitored. When the angle between the $-Z$ axis and the sun is greater than 2 deg, the level detect amplifier then commands the gyros to rate mode for automatic sun acquisition.

The logic for sun occultation/sun acquisition is as follows:

- (1) *Sun occultation.* Loss of sun gate plus sun sensors at null will turn on the gyros in the inertial mode and the spacecraft will stay inertially controlled during the sun occultation period.
- (2) *Sun acquisition.* Loss of sun gate plus sun sensors error not at null will turn on the gyros in the rate mode and the spacecraft will automatically acquire the sun.

A functional block diagram of the sun occultation/sun acquisition logic is shown in Fig. 1.

c. Sun sensor characteristics. There are three types of sun sensors on the *Viking* spacecraft: the sun gate, the acquisition sensors, and the cruise sensors. To mechanize the automatic sun occultation/sun acquisition modes of spacecraft operation, all three sensors are used. The following descriptions of sensor parameters detail the behavior of the sensors in the sun occultation/sun acquisition environment.

Sun gate. In the sun occultation/sun acquisition mechanization, the sun gate will be required to operate in two different modes: (1) to sense the diminishing light intensity of the sun as the sun goes from partial eclipse to total eclipse (occultation mode), and (2) to sense if the angular error between the sun and the $-Z$ axis of

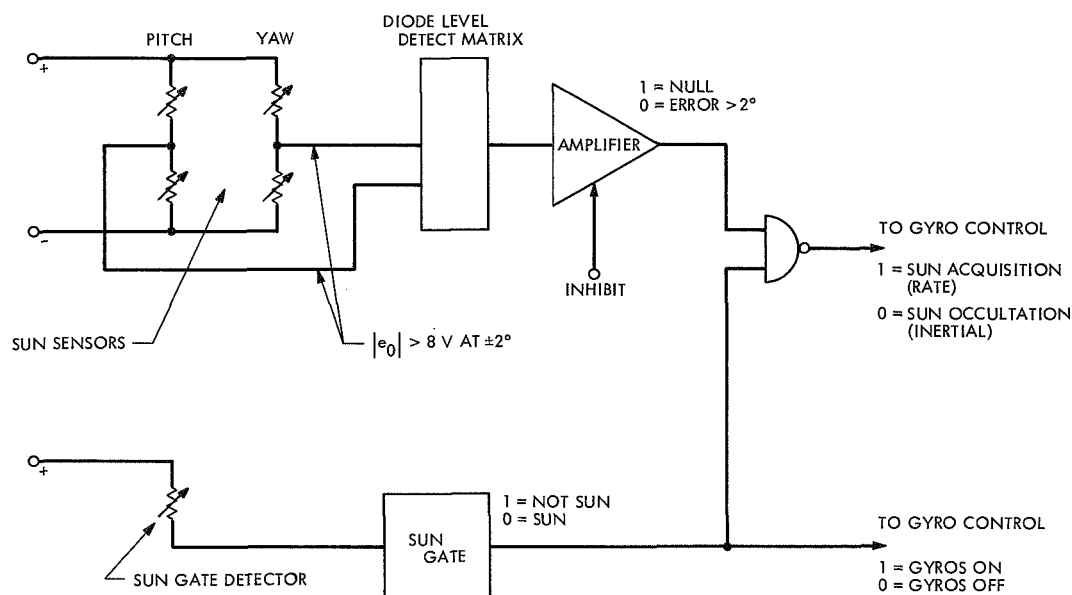


Fig. 1. Viking sun occultation/sun acquisition mechanization

the spacecraft is greater or less than 4.5 deg (acquisition mode).

Table 1 shows the sun gate parameters at Mars (redundant sun gate detectors). It can be seen that the sun gate will not detect sun occultation until the sun intensity has decreased from 5500 (full sun at Mars) to 288 ft-cd (dark threshold of sun gate). This means that sun eclipse will be ~95% complete when the sun gate changes state and turns on the gyros for spacecraft inertial control. Also, the sun gate acquisition threshold is approximately double the dark threshold. This means that the spacecraft will reacquire the sun (after occultation) when ~10% of the sun's surface area is exposed to the sun gate.

Acquisition sun sensors. These sensors cover the full spherical field of view of the spacecraft. The sensors

are arranged in a bridge configuration electrically and are shadowed mechanically such that one light sensitive cell is fully illuminated and one is totally dark during sun acquisition (Fig. 2). The characteristics of the sensor around null are very important for the sun occultation mechanization. Since the outputs of the acquisition sensors and the cruise sensors are summed in the level detect amplifier, it is important that the acquisition sensors are totally shadowed at null. The dark impedance of the acquisition sensors is greater than 20 MΩ. This allows the cruise sensors and the sun gate to be determining elements for the sun occultation mode.

Cruise sun sensors. Figure 3 shows the characteristics of the cruise sun sensors. These sensors have a limited field of view around the -Z axis of the spacecraft (± 20 deg). The voltage scale factor of the cruise sun sensors near null is 5 V/deg. This means that the cruise sensor is voltage-saturated at $\pm 2\frac{1}{2}$ deg from null. This voltage is input to the level detect amplifier and is then used, along with the sun gate signal, to determine the sun acquisition mode. (Gyros are ON in the rate mode.)

Table 1. Sun gate parameters at Mars

Occultation mode		
Threshold	Light intensity, ft-cd	Resistance, KΩ
Acquisition	546	13
Dark	288	18
Acquisition mode		
Threshold	Sun angle, deg	Resistance, KΩ
Acquisition	4.5	13
Dark	4.75	18

An important characteristic of the cruise sun sensors during the period from the start of sun eclipse to sun occultation is the ability of the sensors to track the sun accurately as the sun intensity and size diminish. Tests show the null offset errors to be less than $\frac{1}{2}$ deg, while the sun intensity was reduced from 12,000 to 30 ft-cd. Since the minimum intensity of the sun at the

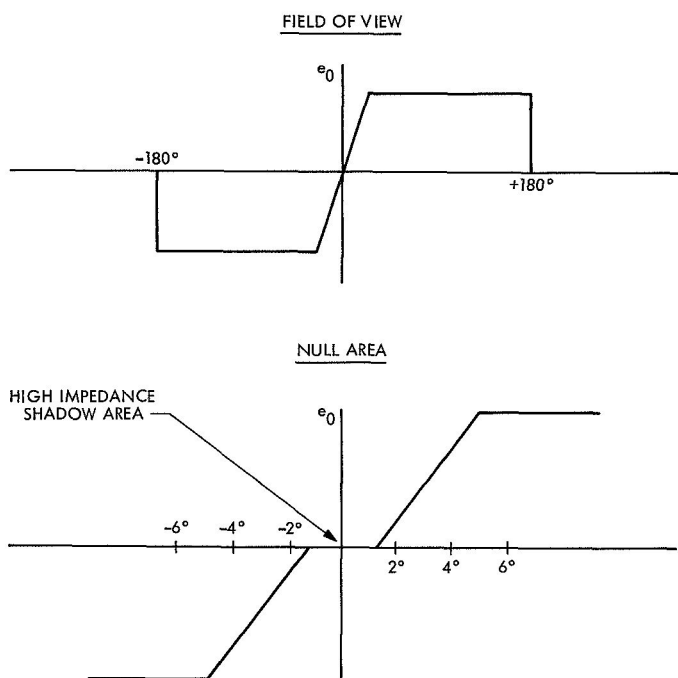
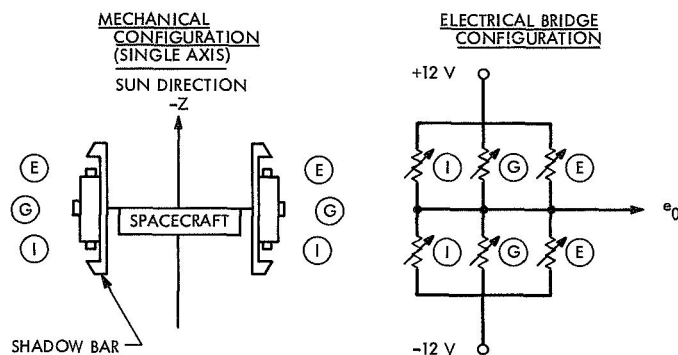


Fig. 2. Acquisition sun sensor

moment of occultation sensing is approximately 300 ft-cd, the sensors are expected to track the sun without significant spacecraft position error.

d. Gyro "inertial start" requirements and performance. A computer program has been generated to show the effects of gyro performance on the spacecraft during the "inertial start" mode of operation. This program uses worst-case start-up transient models in an effort to determine if the spacecraft can accurately maintain its inertial position during the critical gyro spin-motor run-up phase. The results of the computer runs are shown in Figs. 4, 5, and 6.

The graphs show two curves—actual position and pseudoposition. These curves represent spacecraft posi-

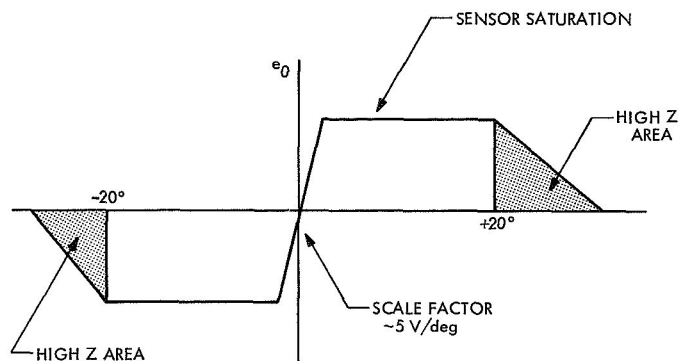


Fig. 3. Cruise sun sensor electrical parameters

tion and gyro integrator position. Input to the program is the gyro rate hang-off during spin-motor run up (4 deg/h), a model of the gyro transfer function versus time, and the gyro spin-motor run-up time. Figures 4, 5, and 6 show the spacecraft behavior for gyro spin-motor run-up times of 1, 3, and 5 min. Typical gyro run-up times at room temperature or above are 60 s or less. The following table summarizes the results:

Run-up time, s	Spacecraft position, deg	Integrator position, deg
60	0.00	0.07
180	0.36	0.08
300	0.71	0.08

Even in the unlikely case of an extremely long gyro run-up, the spacecraft position is held to less than 1 deg. These position errors are small and do not in any way impair the ability of the spacecraft to reacquire celestial references (sun and Canopus) after sun occultation.

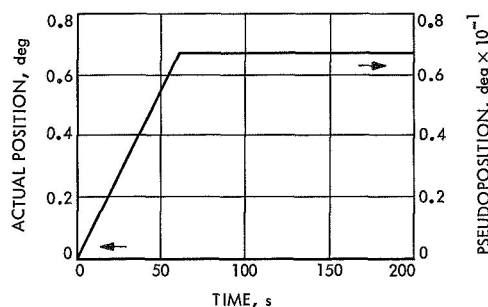


Fig. 4. Actual position and pseudoposition vs time (for 1-min gyro run-up time)

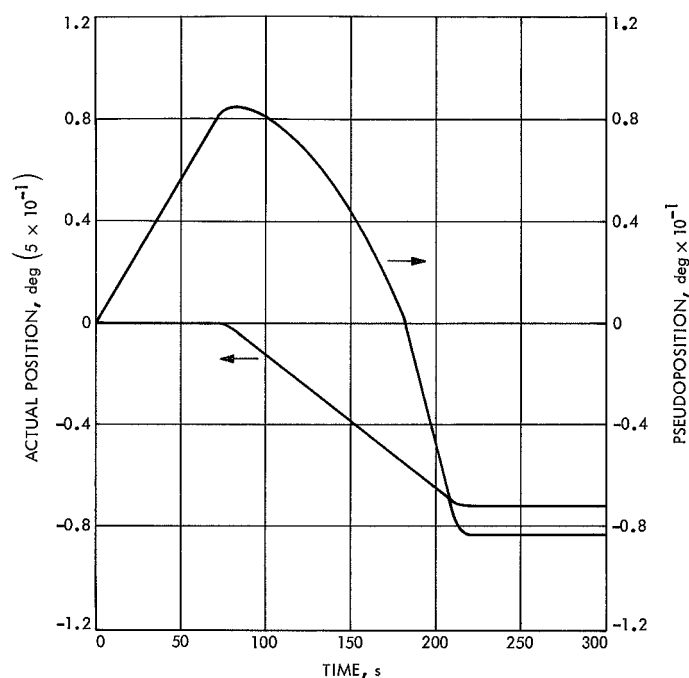


Fig. 5. Actual position and pseudoposition vs time (for 3-min gyro run-up time)

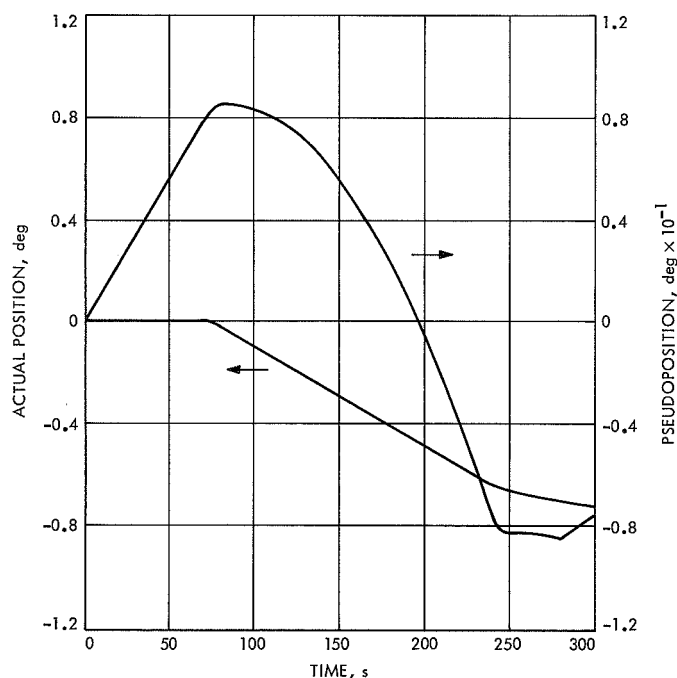


Fig. 6. Actual position and pseudoposition vs time (for 5-min gyro run-up time)

e. Orbital considerations. The sun occultation periods vary in length as the orbital mission progresses. Also, the time between start of eclipse to full eclipse varies

as a function of spacecraft velocity and distance from the planet. Two representative sun occultation periods are listed below:

Parameter	Case 1	Case 2
Distance from planet, km	4,000	20,000
Time after periapsis, min	30	240
Partial eclipse time, s	20	178
Total occultation, min	70	120

f. Conclusions. This article describes the parameters that are relevant to the automatic sun occultation/sun acquisition mode of spacecraft control. This concept is moderately simple to mechanize and has the unique advantage of always being able to detect the difference between the sun occultation mode and the sun acquisition mode.

C. Engineering Mechanics

1. Thermal Design

a. Introduction. Temperature control of the *Viking* orbiter consists of both the design and development of the hardware items, such as louvers, blankets, heaters, etc., and the effective use of these tools to achieve an acceptable overall or system thermal design. This article discusses the thermal design of the *Viking* orbiter from the point of view of the entire system heat balance; this thermal design will later form the basis for the detail design of specific parts of the spacecraft.

b. Temperature control. From a temperature control viewpoint, the *Viking* orbiter (Fig. 7) is composed of three major thermal assemblies plus appendage equipment. The three major elements are: (1) the bus that houses most of the electronic subsystems; (2) the mid-course and orbit insertion propulsion module that consists of tankage and the engine; and (3) the science scan platform that consists of a dual imaging subsystem (two TVs), an infrared thermal mapper, and a Mars atmospheric water detector. Appendage equipments are those items not contained within the three major assemblies, such as the antennas, solar panels, etc.

In relating *Viking* orbiter with past *Mariner* spacecraft, some striking thermal differences become evident. Physical size is the primary difference. The *Viking* spacecraft is considerably larger than the *Mariner* spacecraft, with

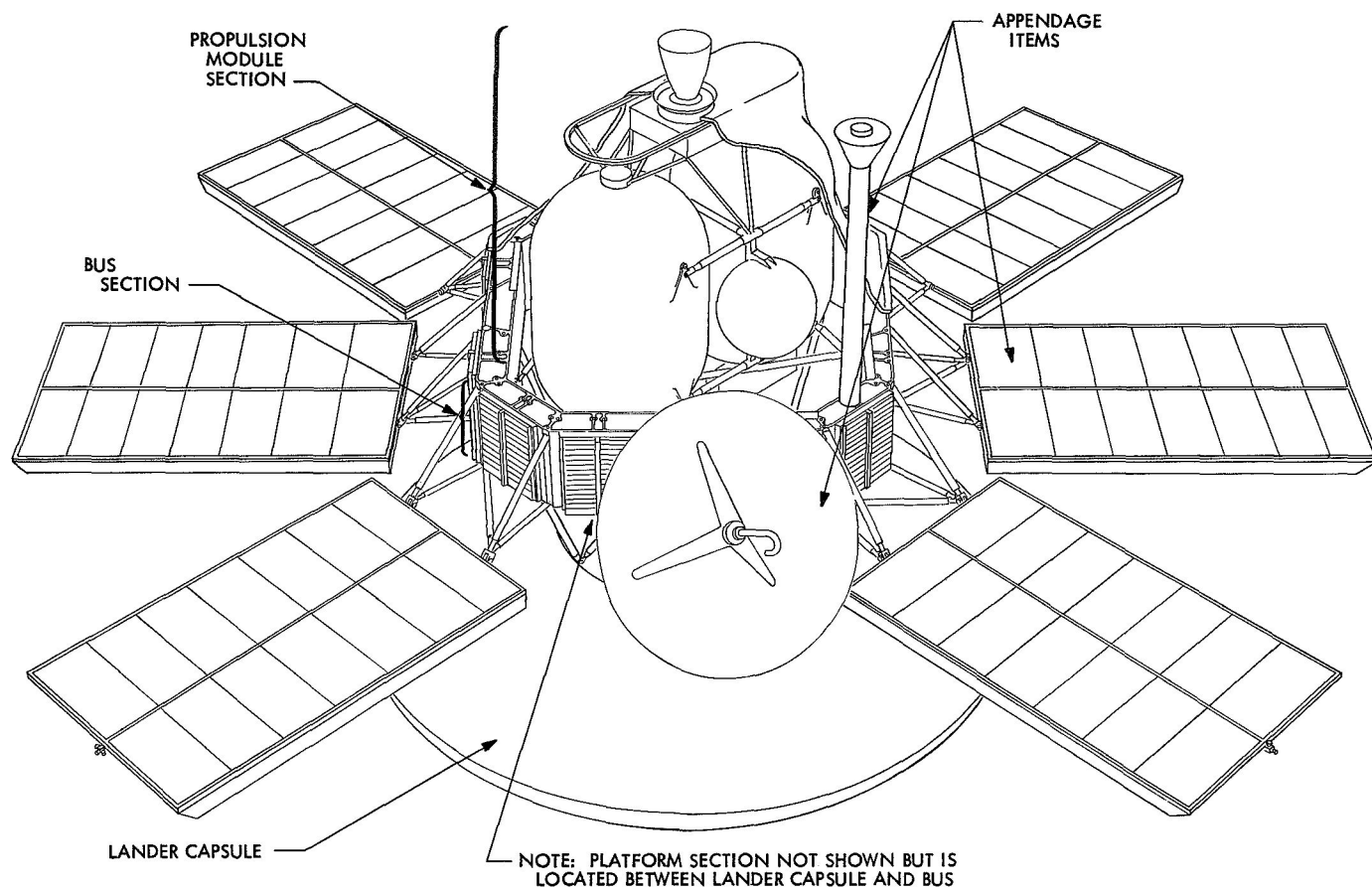


Fig. 7. Viking orbiter simplified configuration

external surface area being roughly three times that of the *Mariner* Mars 1969. On the other hand, internal power dissipation has increased only by a factor of two. Consequently, the conservation of internal power becomes relatively more important. The propulsion module has become much larger; consequently, its temperature control cannot be achieved by thermal coupling to the bus with perhaps a small amount of electrical heating, as was done in the past. The propulsion module must be treated as a separate thermal entity even though roughly one third of its tankage is nested within the bus cavity. This nesting does, in fact, cause a fair amount of coupling; however, because the bus cannot supply a sufficient amount of thermal energy to the propulsion module, the design approach is to negate the coupling by maintaining the propulsion module bulk temperature near that of the bus, creating on the average an adiabatic interface. Solar energy will be utilized to maintain propulsion module temperatures.

The scan platform is presently ill-defined in a specific thermal sense, since mission requirements are still under

study. In a general sense, the *Viking* scan platform is similar to past *Mariner* platforms, with a couple of exceptions. First, the thermal environment is somewhat different due to the large obstruction to viewing of space created by the *Viking* lander; and second, the complement of instruments are new, with perhaps tighter thermal constraints. Even with these differences, the temperature control design approach to the *Viking* platform remains essentially the same as for *Mariner* spacecraft.

Since very little thermal design has been done on appendage items, they will not be discussed here.

c. Bus. Other than the striking increase in size, the *Viking* orbiter bus is generally similar to those of past *Mariner* spacecraft. It appears that past techniques will be adequate and will offer a fair degree of design flexibility and operational reliability. These techniques are of either the passive or semi-active type and include painted and polished metal surfaces, multilayer insulation, electrical heaters, and louver arrays.

The *Viking* bus contains 16 equipment bays. Of the 16 bays, 12 contain electronic subassemblies, two contain the attitude control gas bottles, one contains the Canopus sensor, and one serves as a mounting platform for the high-gain antenna. The shape is an imperfect octagon with four sides that contain three bays each and four that contain one bay each. The bays form a ring that encloses about a third of the propulsion module. The central cavity is to be closed off on one end by a multi-layer blanket installed as a "drum skin" over the cavity to the outer periphery of the octagon. Out of the other end of the bus, the protruding propulsion module will also be blanketed, thus forming an enclosure that includes the bus cavity and propulsion module.

At present, four of the 12 bays that house electronic subsystems are nominally dormant during cruise. The temperatures of these four bays will be maintained by four 10-W substitution heaters. The four zero- or low-power bays that contain the attitude control gas bottles, Canopus tracker, and the high-gain antenna mount will be sandwiched between dissipating bays, thus obtaining thermal energy by conduction and radiation coupling.

Further details regarding power and power distribution within the bus are contained in SPS 37-62, Vol. I, pp. 35-43.

d. Propulsion module. Thermal efforts relative to the propulsion module concentrated on two problem areas: (1) maintaining the module warm during cruise and in Mars orbit, and (2) keeping various components within acceptable temperature limits during the long orbit insertion burn. Both of these problems are more severe for *Viking* 1975 than for the similar but smaller *Mariner* Mars 1971 spacecraft. Solution of the first problem is based on the use of a new development item, dubbed the solar energy controller (SEC). A feasibility model of the SEC has already been tested (SPS 37-62, Vol. I), and the first thermal flight prototype is currently being fabricated. Standard techniques, such as selective thermal coating, insulating, etc., hopefully will be sufficient for solving the soak-back heating that follows the orbit insertion burn.

Solar energy controller. Early calculations indicated that the temperature control of the propulsion module would require 20 to 30 W more electrical power than *Mariner* Mars 1971. This additional energy requirement results from a larger propulsion module (a larger radiating area), an assumed low thermal coupling with the bus,

and a minimum solar intensity at Mars aphelion (rather than the near perihelion condition for the *Mariner* Mars 1971 spacecraft). Obtaining 30 more watts by additional solar panel area would increase the spacecraft weight by at least 8 lb. For this same weight, 80 W can be provided to the propulsion module with four SEC units.

Briefly, the SEC is a shuttered greenhouse window. The greenhouse window is transparent to the solar energy allowed through the shutters and opaque to the infrared energy emitted from within the orbiter. For the flight model, the plane of the window is approximately in line with the sun. Articulated louvers normal to the sun regulate solar energy onto a highly reflective surface. The energy reflected from the surface passes through the window and is reflectively spread onto the propulsion tanks, structure, etc. By being independently controlled from the ground through the flight command subsystem, the SEC will not only maintain the propulsion module at a desired average temperature, but also minimize the thermal gradients expected in the propulsion module.

Soak-back heating. Except for the engine skirt, the *Mariner* Mars 1971 engine is the same as the *Viking* 1975 engine, and keeping cognizant of the *Mariner* Mars 1971 thermal control techniques has been the principal effort in this region of the propulsion module. Without a major modification, it is doubtful that any significant improvement can be made over the temperature control methods established by *Mariner* Mars 1971. For example, the addition of a louver system for dumping heat in the immediate vicinity of the engine or the development of a heat pipe arrangement to transport heat to the top of the propulsion module are possible, but not currently planned. Temperatures higher than those determined for *Mariner* Mars 1971 can be expected because of the significantly longer *Viking* 1975 orbit insertion burn.

Early calculations indicated the propellant tanks would reach a temperature locally higher than the design point of 90°F, due to the longer burn time. The *Mariner* Mars 1971 thermal control model test results have also confirmed that a temperature above 100°F should be expected. As soon as the 1971 test results become available, the propulsion analytical model will be refined and how much the local tank temperatures will rise above 100°F can be estimated. This design point has been tentatively increased to 120°F, and the mounting of the motor has been changed from that of a magnesium bridge to a lower conductive truss support.

e. Scan platform. The scan platform is composed of advanced new scientific instruments. The most thermally sensitive of these instruments are the Mars atmospheric water detector (MAWD) and the optics of the two TV telescopes. These instruments cannot tolerate temperature gradients of over 3°F across the longest dimension, and the absolute temperatures must be maintained to about the same degree of accuracy.

The scan platform internal thermal environment is stabilized by the application of 60 W of continuously applied electrical power. The 60-W value is preliminary and is based on the assumption that the total thermal power required for temperature control will not be significantly different than that required for past *Mariner* spacecraft. The 60 W will be derived from electrical heaters operated off the main regulated dc bus when the science instruments are off. For the science on condition, the 60 W will be derived from a combination of electronic dissipation and heater dissipation. The essential point is that a regulated 60 W will be dissipated and that this power will be dissipated in thermally identical locations (to the extent possible) regardless of whether the instruments are on or off.

The powering of all platform heaters from a regulated supply rather than off the unregulated supply is a new departure and derives from the desire to achieve a constant power input to the platform. The unregulated supply voltage can vary from 28 V to perhaps 50 V, depending upon mode (battery or solar panels) and solar panel temperatures. These large power variations lead to significant amounts of test time to establish design adequacy under all conditions and also may be in conflict with the temperature constraints. The regulated supply will eliminate some of these problems.

It is anticipated that all of the scan platform will be enclosed in a multilayer insulation blanket to conserve thermal power and thermally integrate the instruments. Also, the platform will be thermally isolated from the bus and sun. To compensate for uncertainties in insulation performance and to prevent a cascading effect should any single instrument fail, some type of variable emittance device will be utilized, probably a louver array.

Externally, the scan platform is influenced by the interaction of the backside of the solar panels, the bus interface, and the large biocanister of the *Viking* lander, which presents one effect when attached and another

effect when jettisoned. It is believed that the insulation blanket will tend to dampen these effects.

In Mars orbit, the planet influences the platform to a predictable but significant degree. All of the requirements and constraints for the scan platform are not yet entirely known; but the scan platform instruments, particularly the MAWD, must view wide areas of Mars to aid in the selection of an appropriate landing site.

The MAWD instrument is the most difficult instrument on the scan platform from a thermal control design standpoint since it must have not only low thermal gradients but a low (−100°F) temperature for the detector. This low temperature is provided by a thermal radiator plate, which is part of the MAWD, that views out from the scan platform blanket.

The performance of the MAWD radiator is influenced externally by all of the surfaces within view, which include the lander, bus interface, and solar panel backside. The view factors to these surfaces vary with both cone and clock angle of the scan platform, so many positions must be evaluated to determine the worst case. Also, conditions after the lander is jettisoned must also be considered as well as the transient thermal effects of the Mars infrared and reflected solar energy. The internal heat load to the MAWD detector consists of radiation from the instrument walls, thermal conduction through the detector mounting system, and the radiant flux received while viewing the planet.

The temperature control goals for the MAWD are dictated not only by the ambient temperature (70°F) and gradient requirements (3°F max) but, in addition, by the detector temperature level (−100°F) and the allowable transient temperature change rate (3°F/h). These temperature requirements are extreme and to achieve them a very careful heat balance will be necessary and, perhaps, new and unique thermal control devices will have to be considered.

2. Ground Handling and Assembly Equipment

a. Introduction. Engineering mechanics effort for the *Viking* 1975 Project includes the design, fabrication, and, in many cases, the use of the mechanical ground handling and assembly equipment. Although many of these activities are still in the planning stage, the basic ground handling philosophy has been selected and general techniques have been outlined for the major activities. This article briefly discusses the ground handling

design approach, assembly and test sequences and equipment, and indicates departures from previous practices. Although handling of the entire *Viking* spacecraft is not JPL responsibility, combined operations which involve the orbiter and other major hardware items must be considered.

b. Design approach. The basic ground handling philosophy considers the operations at the Air Force Eastern Test Range as the most important. This does not detract from the importance of a well-designed ground handling system for the months of preflight testing, but emphasizes the need for a safe and reliable final assembly. To this end, the evolution of the basic equipment and procedures will begin in the early orbiter development tests and during the assembly and checkout of the proof test model and flight systems at JPL. For tests or operations which are planned for the orbiter development phases but are not part of the Cape Kennedy operations, the prime concern will be for the safety of the spacecraft and the operating personnel.

In past flight programs, the spacecraft was shipped to Cape Kennedy in an as completely assembled condition as possible because this method supported the various assemblies in the preferred way, reduced the number of containers and lost equipment, and reduced the packing and unpacking time. Shipments were made in standard-size electronic vans with approved suspension systems. Certain equipments, such as gyros, solar panels, batteries, etc., had peculiar characteristics and were handled separately. While this approach will still be followed where possible with the *Viking* orbiter, the large physical size of the assembled orbiter precludes using this technique directly.

The orbiter consists of logical groupings of equipment which have similar if not identical assembly and qualification sequences. These major groups are the bus equipment, the propulsion module, the adapters, solar panels, and possibly the science platform. Separately, these groups (as well as the ground equipment to handle them) can be transported in standard-size vans, and to that extent the shipments will be made as major assemblies.

The bus, adapters, and propulsion module must be assembled as a base for the mounting of the accessorial assemblies. These assemblies will at least include the solar panels and the thermal blankets but may include the high-gain antenna, the science instruments, the attitude-control gas systems, and some electronic equip-

ment. An attempt will be made during the detail design period to keep the number of items which require late assembly to a minimum.

c. Assembly sequences. The basic assembly procedure permits the easy replacement of the major groups (lander/capsule, propulsion module, and bus) in the event a quick change is necessary during final checkout or countdown. Also, the propulsion module and the bus electronics have significantly different pre-assembly checkout sequences (as is the case with the lander/capsule) and can be verified separately without imposing undue constraints on the other systems. By defining these criteria early in the design of the equipment, it is possible to have a greater influence on making the interface hardware durable and easy to assemble. This is true not only for the major

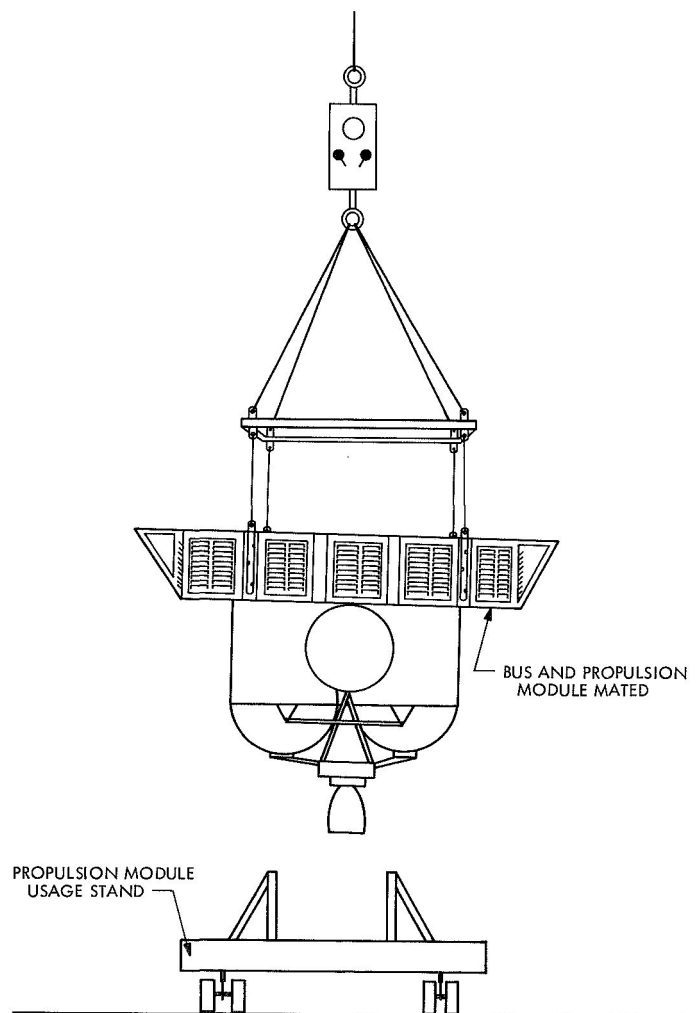


Fig. 8. Orbiter bus to propulsion module assembly operation

groups but for the appendage equipment, which must be attached late in the assembly sequence.

The final flight assembly sequence of the orbiter includes testing, calibration, or mechanical aligning of flight-qualified assemblies and a checkout as a system. During this period, all systems are in a most hazardous condition: propellants are loaded, pressure vessels are charged, and ordnance devices are installed.

Figure 8 illustrates the mating of the orbiter bus to the propulsion module. Figure 9 illustrates the basic structural components of the orbiter.

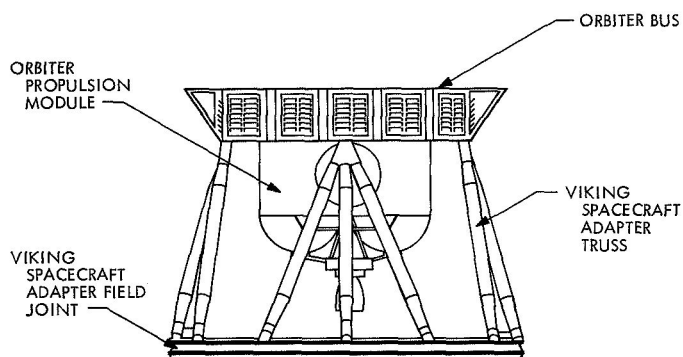


Fig. 9. Basic orbiter structural components

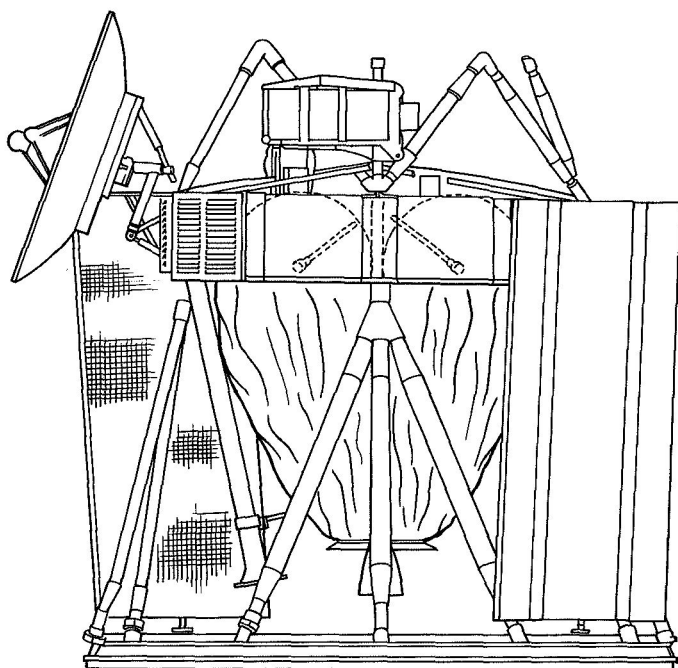
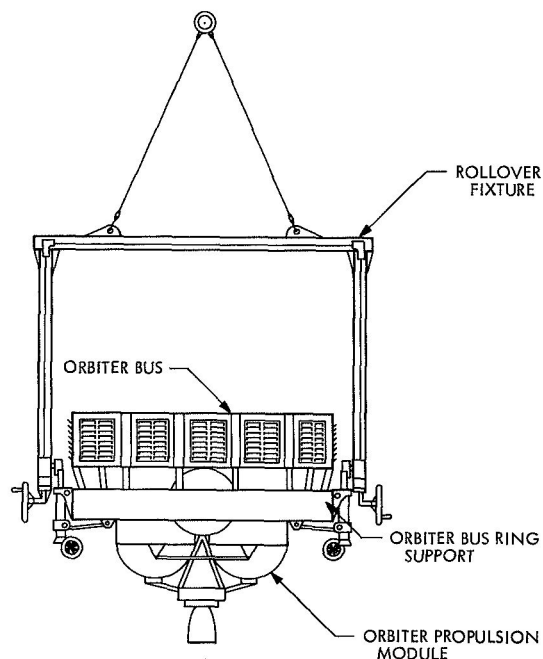


Fig. 10. Viking orbiter launch mode configuration (solar panels removed)

Upon completion of final assembly (Fig. 10), the orbiter is delivered to the Martin-Marietta Corp. facilities for mating with the lander/capsule. When spacecraft final assembly and testing is completed, the spacecraft is encapsulated in the nose fairing and delivered to the launch pad for mating to the launch vehicle.

d. Major test sequences. In the JPL space simulator test, the simulated solar source shines vertically from above. Because the cruise orientation is 180 deg from the launch orientation, this places the orbiter in an upside-

(a) STEP 1



(b) STEP 2

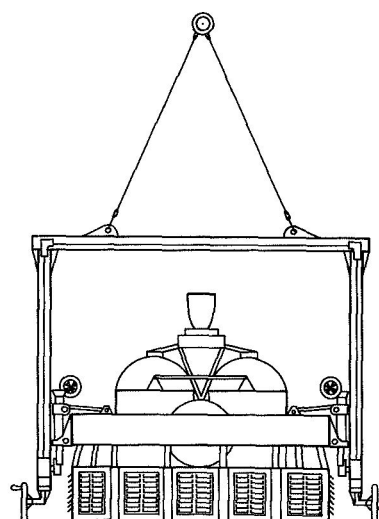


Fig. 11. Orbiter Z-axis roll-over 180-deg operation

down test configuration. This operation of turning the spacecraft upside down and suspending it in a vacuum chamber was previously accomplished with the *Mariner Venus 67* spacecraft; however, that spacecraft was considerably smaller than the *Viking* orbiter.

To orient the orbiter to the overhead light source, the bus (with support ring attached) is assembled to the propulsion module, which is installed in the usage stand (Fig. 8). After the roll-over fixture is attached to the bus support ring, the load is transferred to the roll-over fixture. The rotation of the assembly is accomplished by using a 50 to 1 ratio drivewheel until the 180-deg position is obtained (Fig. 11). When this operation is completed, the assembly is transferred from the roll-over fixture to a work dolly where additional assembly and testing is performed prior to installation in the simulator chamber. This sequence will first be accomplished by using the orbiter thermal control model and later the proof test model as a means of assuring a safe and well-checked technique for the flight orbiter. There is no plan to include the flight capsules in this sequence, although the option is available and will be verified during proof-test-model combined testing with the lander/capsule.

For electrical test and checkout of the orbiter in the JPL Spacecraft Assembly Facility, the system ground support equipment, sensors, stimuli, etc., are used. The systems test stand, which is used to assemble the orbiter (Fig. 12), is a portion of the systems test complex. During the testing period, the load-carrying shear plates for the electronic assemblies may be removed for troubleshooting tests. To provide support and prevent structural damage, a nonflight support is installed between the aft ring of the bus and the systems test stand. This ring makes it possible to test in a partially disassembled condition while still maintaining the structural integrity of the orbiter bus structure.

The systems test stand has a Z-axis roll and X- or Y-axis pitch capability. This is required for some of the alignment and autopilot checks. The rates of movement are compatible with the orbiter power on, power off, condition constraints.

e. Other test sequences and equipment. Some of the ground handling equipment required for other test sequences is listed in Table 2. Some of this equipment will

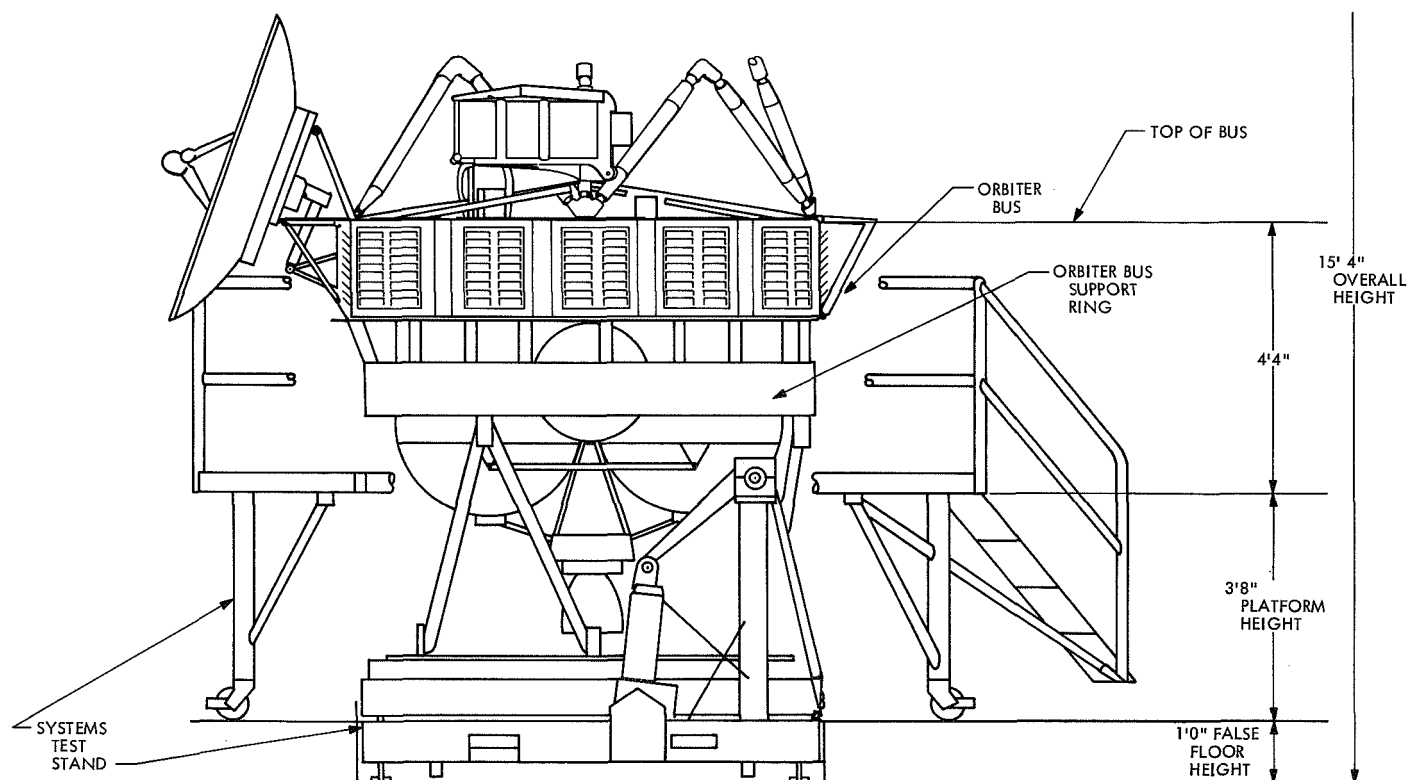


Fig. 12. Viking orbiter installed in systems test stand

Table 2. Major assembly, handling, and shipping equipment

Item	Function	Configuration	Vehicle
Orbiter Z-axis rollover assembly	To rotate Z-axis of orbiter 180 deg for proper orientation for thermal control test	Bus and propulsion module; when tested, cruise mode	Thermal control model, proof test model, flight orbiters 1 and 2
Systems test stand	For all systems testing	Bus and propulsion module plus lander	Proof test model, flight orbiters 1 and 2
Transport trailer/cover	For transport of components; always component level of assembly (never an assembled vehicle)	Bus to a given level of assembly; propulsion module in usage stand	Dynamic test model, thermal control model, proof test model, flight orbiters 1 and 2
Lifting fixture	Vertical lift operations or relocation	Bus alone to launch mode assembly	Thermal control model, dynamic test model, proof test model, flight orbiters 1 and 2
Solar panel transport trailer	Storage, transport from vendor to test site, to Air Force Eastern Test Range	Solar panel mounted in handling frame, installed in trailer (3 panels per trailer)	Dynamic test model, proof test model, flight orbiters 1 and 2
Bus, storage, and service dolly	Support bus for assembly operations	Bus and installed systems	Configuration mockup, thermal control model, dynamic test model, proof test model, flight orbiters 1 and 2
Orbiter bus ring support	Interfaces to —Z ring of bus for structural support in all modes of testing	Structure between bus and support structure (service dolly, systems test stand, transporter, etc.)	Dynamic test model, thermal control model, proof test model, flight orbiters 1 and 2
Weight and CG location fixture	To determine from the Z-axis, along the X- or Y-axis, the CG migration	Launch or cruise mode	Proof test model, flight orbiters 1 and 2
Electronic assembly installation tool	Installation of the electronic assemblies to the bus to prevent damage to components and cabling	Attaches to 14 of the 16 vertical longerons of the bus; sliding drawer concept	Proof test model, flight orbiters 1 and 2
Suspension and installation equipment for 25-ft simulator	Suspension of orbiter and lander —Z-axis vertical in 25-ft space simulator chamber	Early propulsion temperature control test configuration to Viking spacecraft test configuration	Thermal control model, proof test model, flight orbiters 1 and 2

affect flight hardware design; however, throughout developmental testing, two main constraints were required:

- (1) The ground handling equipment must not impart loads to the vehicle greater than the launch environment.
- (2) Alternate mechanical attach points must be provided for the ground handling equipment to the orbiter, thus reducing the design constraints and physical wear to the flight accessorial mounting points.

Based on the operations flow charts, it is apparent that test models experience seven or more assembly recycle operations to establish the required test configuration. The flight vehicles are subject to four recycles, with the fourth being the final flight assembly operation.

Some of the other support equipment required includes the support for the antenna test model, the support for configuration mock-up vehicles, the solar panel handling, storage and shipping trailers, and many accessorial storage and shipping containers.